EA303 WIND TUNNEL
EXPERIMENT VI

PREDICTING BEECHCRAFT BONANZA AIRCRAFT PERFORMANCE FROM WIND TUNNEL TESTS

I. Purpose

1. To experimentally determine the drag polar for the aircraft model.
2. To experimentally determine maximum lift coefficient, slope of the lift curve, and minimum drag coefficient for the aircraft model.
3. To predict maximum lift coefficient for the full scale aircraft using appropriate Reynolds number corrections to the model experimental data.
4. To construct a predicted drag polar for the full scale aircraft using appropriate Reynolds number corrections to the experiment data.

II. References

1. Pope, *Low Speed Wind Tunnel Testing*, Chapter 7.
2. Dwinnell, *Principles of Aerodynamics*, sec 7.9
3. Hurt, *Aerodynamics for Naval Aviators*, pp. 59–61.

III. Introduction

Aircraft performance analysis critically depends on an accurate knowledge of the aircraft drag polar. Preliminary estimation of the drag polar is generally made analytically. However, there are limitations to analytical estimates. Hence, it is common practice to test a scale model of the proposed aircraft in a wind tunnel or tunnels in order to obtain more accurate information on the drag polar. Since wind tunnels are generally of insufficient size to accept full size models due to tunnel and model costs, and since it is generally more important to match the full scale velocity during the test, the full scale Reynolds number is not matched. Thus, it is necessary to correct for the effects of the reduced Reynolds number experienced by the model. These effects are called Reynolds number or "scale" effects.

Correcting wind tunnel data for Reynolds number effects is a difficult empirical task because of the correlation between full scale aircraft and models. Hence, the aircraft designer is forced to use approximate methods.

IV. Theory

Reynolds number or scale effects are closely associated with the character of the boundary layer. Recall that the boundary layer may be either laminar or turbulent, and that transition from a laminar to a turbulent boundary layer is determined by such factors as Reynolds number, pressure gradient, initial airstream turbulence, surface roughness, etc. A turbulent boundary layer has a velocity profile that is considerably steeper, i.e., has more energy near the surface than a laminar boundary layer (see Fig. 6–1). Thus, a turbulent boundary layer causes more skin friction drag than a laminar boundary layer. In addition, since the turbulent boundary layer has more energy near the surface it is better able to overcome an adverse pressure gradient (pressure increases in the direction of the flow) and hence delay separation. Separation is the detachment of the flow from the surface.

In order to illustrate these ideas and the importance of Reynolds number effects, consider an airfoil as shown in Fig. 6–2. Initially (Fig. 6–2a) the Reynolds number is sufficiently low that natural transition to a turbulent boundary layer does not take place. Because the laminar boundary layer cannot penetrate the rising pressure gradient that occurs after the minimum pressure point, laminar separation takes place, with its large associated viscous wake. The drag is quite large.

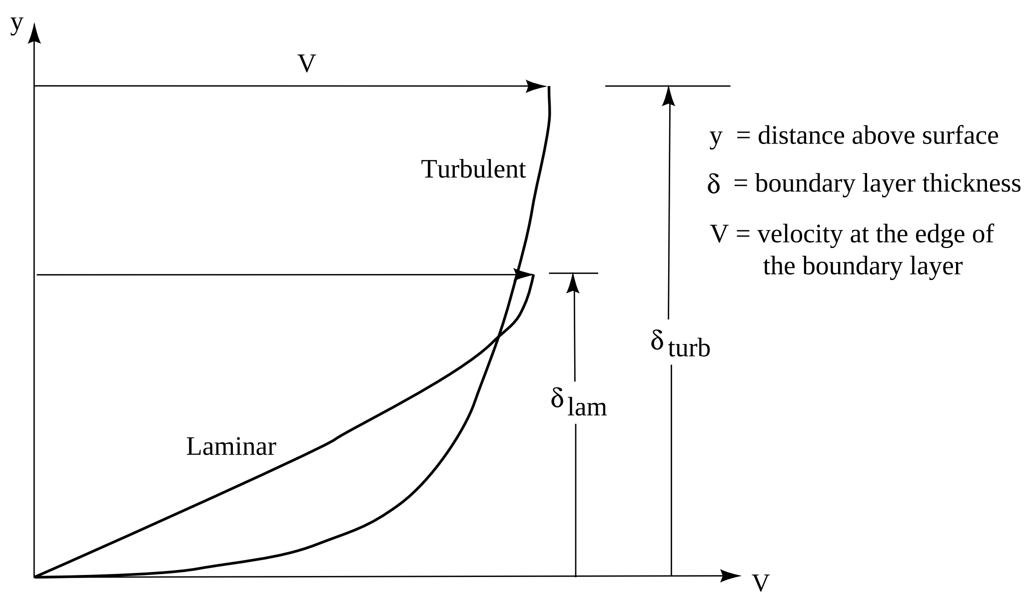

Figure 6–1. Boundary layer velocity profiles.

Figure 6–2b illustrates the situation where the flow Reynolds number is sufficiently large that natural transition to a turbulent boundary layer takes place. The turbulent boundary layer is better able to penetrate the adverse pressure gradient, separation is delayed, the extent of the viscous wake is decreased and the overall drag decreased.

Figure 6–2c illustrates that as the Reynolds number is further increased the natural transition point moves forward. Hence, there is a greater proportion of turbulent boundary layer and the drag is increased with respect to that for the case illustrated in Fig. 6–2b.

V. Scale Effect on Lift Curve

The scale effects on the lift curve may be summarized for NACA 4 and 5 digit series airfoils as follows:

a. The effects on the lift curve slope are small. However, the lift curve slope will become more linear for low angles of attack as Reynolds number increases. The lift curve slope also increases slightly.

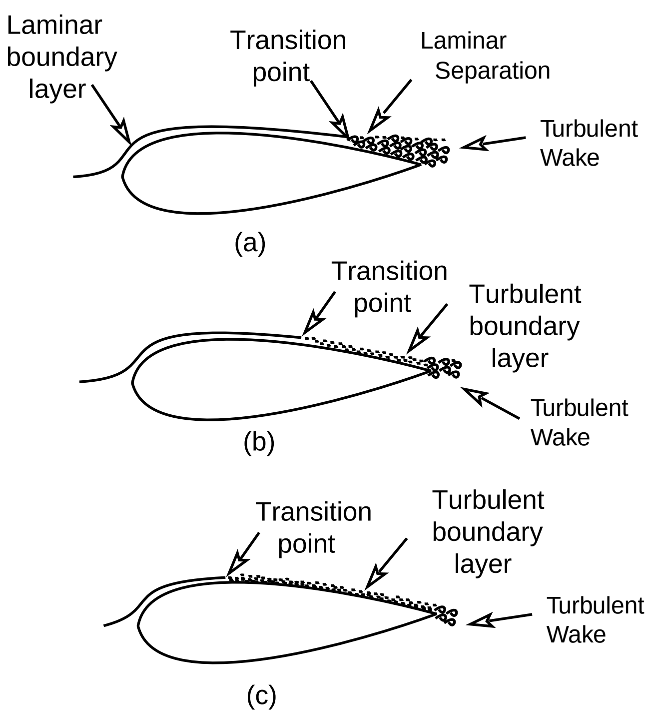

Figure 6–2. Effect of increasing Reynolds number on the boundary layer flow.

b. The stall becomes more abrupt (see Fig. 6–3).

c. $C_{l_{max}}$ and $\alpha_{C_{l_{\max}}}$ increase (see Fig. 6–3).

The lift curve for the full scale aircraft can be estimated from the model lift curve using the following procedure (see Fig. 6–4).

1. Estimate the full scale aircraft maximum lift coefficient by direct ratio, using Fig. 6–5. A proportional relationship can then be written

$$\frac{(C_{L_{\max}})_{\mathrm{ac}}}{(C_{L\,\max})_{\mathrm{model}}} = \frac{(C_{L\,\max})_{\mathrm{wing}}}{(C_{L\,\max})_{\mathrm{wing}}} \quad \frac{(@\mathrm{Re})_{\mathrm{ac}}}{(@\mathrm{Re})_{\mathrm{model}}}$$

The Reynolds number is based on the mean aerodynamic chord of the model and of the aircraft. Recall that the mean aerodynamic chord, MAC, is given by

$$MAC = \frac{2}{3}\left(c_t + c_r - \frac{c_t c_r}{c_t + c_r}\right)$$

for straight tapered wings where c_t is the chord at the tip and c_r is the chord at the root.

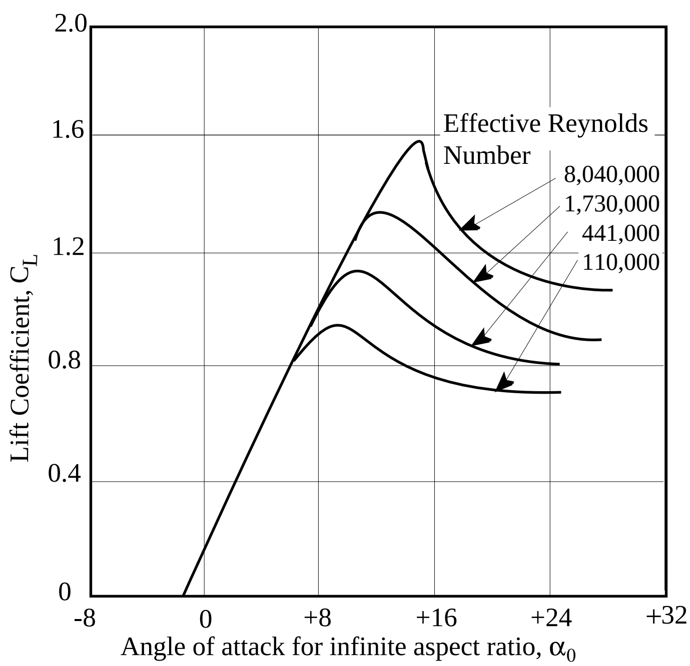

Figure 6–3. Effect of Reynolds number on the lift curve.

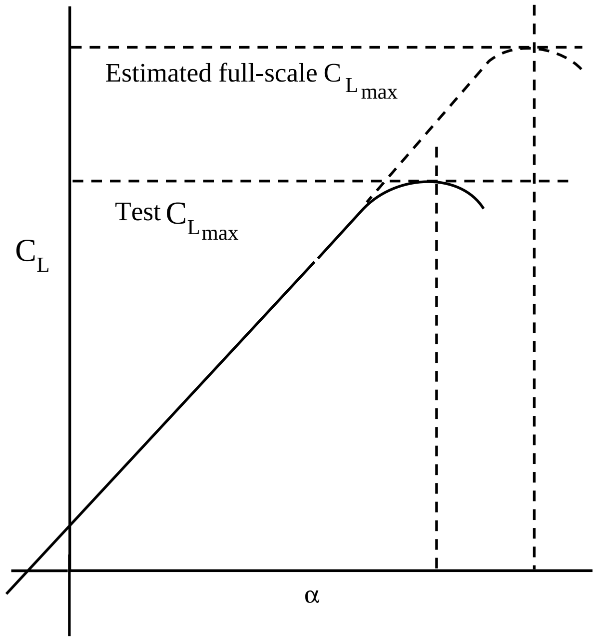

Figure 6–4. Construction of full scale lift curve from test data.

Solving for $(C_{L\,\max})_{\mathrm{ac}}$ yields

$$(C_{L\,\max})_{\mathrm{ac}} = (C_{L\,\max})_{\mathrm{model}} \frac{(C_{L\,\max})_{\mathrm{wing}}}{(C_{L\,\max})_{\mathrm{wing}}} \quad \frac{@(\mathrm{Re})_{\mathrm{ac}}}{@(\mathrm{Re})_{\mathrm{model}}} \tag{6 − 1}$$

Draw a horizontal line on the C_L vs α plot at this value of $C_{L\,\max}$.

2. Extend the linear portion of the model C_L vs α curve with the same slope, a.
3. Raise the nonlinear portion of the model C_L vs α curve until it has the proper $C_{L\,\max}$ value. Shift it laterally until it joins the linear portion of the scale lift curve.

This procedure results in a full-scale lift curve having the proper value of $\alpha_{0\mathrm{L}}$, slope, a, and $C_{L_{\max}}$, but with $\alpha_{C_{L_{\max}}}$ too large and a stall that is too gentle .

VI. Scale Effect on Drag

The determination of the scale effects on the drag are more complicated than for lift. The more important factors which affect the drag are:

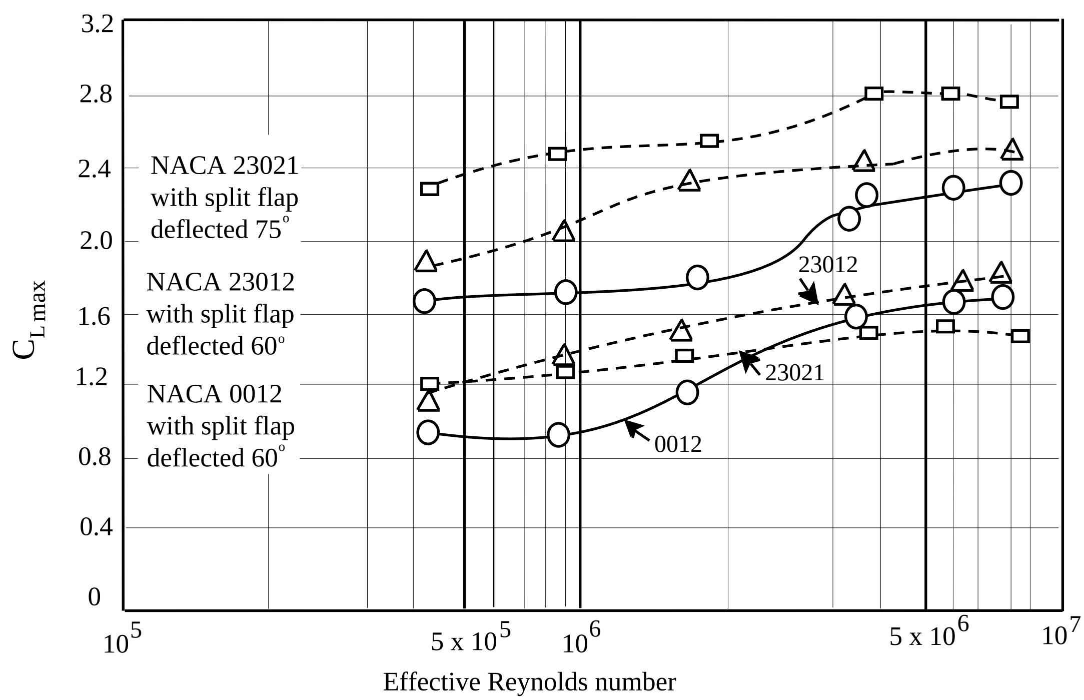

Figure 6–5. Variation in $C_{L_{\max}}$ with Reynolds number; general airfoil trend.

a. The scale effect on the airplane efficiency factor or Oswald efficiency factor: this effect is associated with the change in lift distribution with Reynolds number. Fortunately, experimental results indicate that the effect of Reynolds number is negligible. Hence wind tunnel test may be used to determine the full scale efficiency factor.

b. The reduction of minimum drag coefficient due to increased Reynolds number.

c. The increase of minimum drag coefficient not apparent in wind tunnel measurements, due to the fact that all small protrusions, manufacturing irregularities and other excrescences of the full scale aircraft are not included on the 'scale' model. It is difficult to determine whether these two factors (decrease and increase in drag) compensate for each other. For simplicity we assume that they do; therefore, by assumption

$$(C_{\mathrm{D}_{0_{\min}}})_{\mathrm{ac}} = (C_{\mathrm{D}_{0_{\min}}})_{\mathrm{model}} \tag{6-2}$$

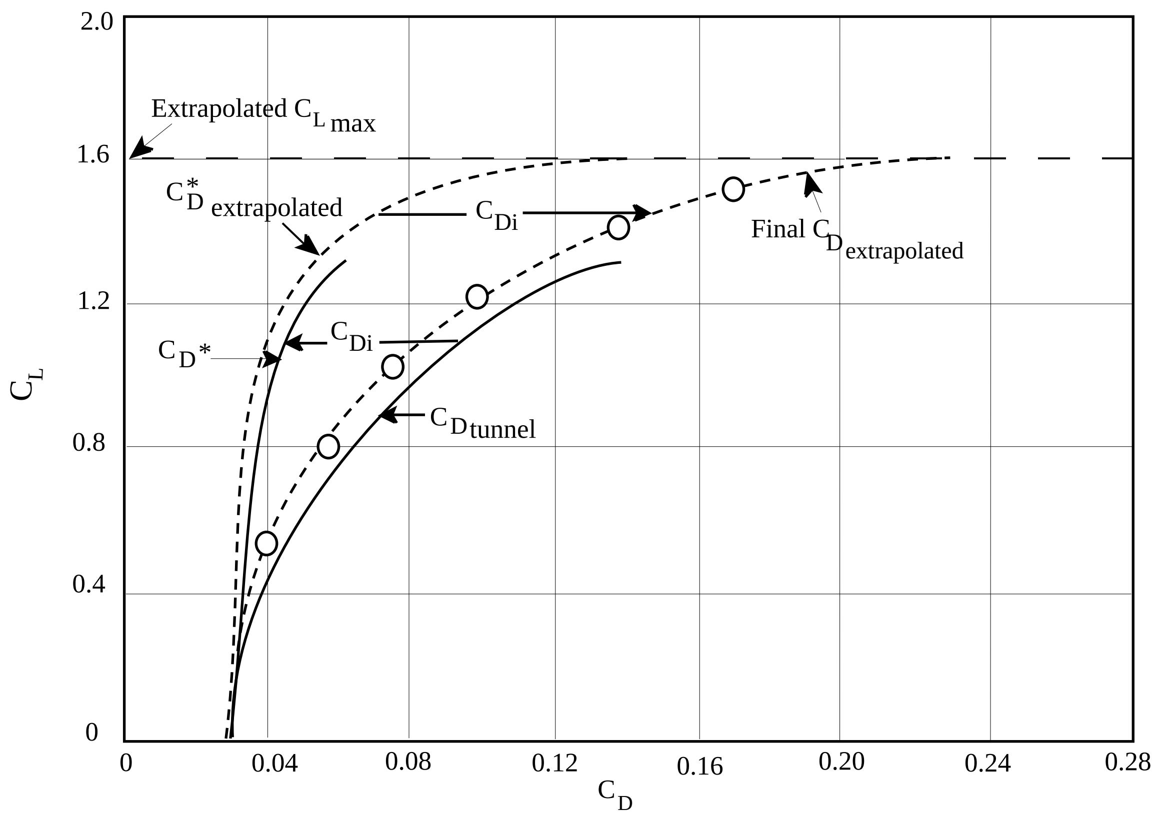

Figure 6–6. Extrapolating airplane drag curve to full-scale Reynolds number.

To extrapolate intermediate points of the model drag polar to points of the aircraft drag polar, the following method (see Fig. 6–6) can be used.

1. Plot the tunnel model C_L versus C_D curve.
2. At various points along the curve, subtract $C_L^2/\pi A\!\!R$ and plot the remaining drag coefficient using

$$C_D^* = C_{D_{\text{tunnel}}} - \frac{C_L^2}{\pi A\!\!R} \qquad (6-3)$$

   The aspect ratio for the model is based on the planform of the wing. The efficiency of the wing in the above equation is assumed to be one.
3. Estimate $C_{L\,\max}$ from Fig. 6–5 and Eq. (6–1). Extend the resulting maximum C_D^* vs C_L curve to be tangent at the horizontal line at the predicted $C_{L\,\max}$

of the aircraft. The increased curvature of C_D^* should be moved as described in the section on lift curve effects.

4. Using the new aircraft, C_L, calculate $C_L^2/\pi AR$ and add it to the extrapolated C_D^* to get the extrapolated aircraft drag coefficient curve. The resultant $C_{D_{\mathrm{ac}}}$ curve will be identical to the original curve at low values of C_L, but a new curve will result at high values of C_L.

VII. Physical Setup

The platform strut type balance is used to support a 1/16 scale model of a Beechcraft Bonanza F33A aircraft in the blue wind tunnel. In this experiment, the balance is used to measure lift, drag and moment by rotating the model in the pitch plane. The thermometer, barometer and inclined manometer are used to establish the flow properties in the usual manner.

VIII. Procedure

1. Perform an auto zero.
2. Before starting the wind tunnel, obtain tare readings of lift, drag and moment at angles of attack from $-6°$ to $16°$.
3. Record lift, drag and moment at an inclined manometer reading of 7 inches of alcohol and at angles of attack as used in number 2 above.
4. Calculate lift, drag and moment coefficients for the model.
5. Plot the following parameters versus angle of attack on separate graphs:
   a. lift coefficient C_L vs α;
   b. drag coefficient C_D vs. C_L;
   c. $C_{M_{TR}}$ vs. C_L.
6. Using the scaling techniques discussed above, as appropriate, find:
   a. $(C_{L_{\max}})_{\mathrm{ac}}$, $(\alpha_{C_{l_{\max}}})_{\mathrm{ac}}$, a_{ac};
   b. plot the aircraft drag polar and $C_{L_{\mathrm{ac}}}$ vs α.
7. Fill in the table of results and include complete sample calculations, and show other pertinent work (e.g., how e_{ac} is obtained).
8. Discuss results, errors, unusual conditions if any, etc.

IX. Table of Results

	Wind Tunnel Model	Full Scale Aircraft
$C_{L\,\max}$		
$\alpha_{C_{L\max}}$		
a		
α_{0L}		
$C_{D\,\min}$		
e		
C_{mac}		

MODEL DATA:

$C_r = 5.5''$ (chord at wing root)

$c_t = 2.75''$ (chord at wing tip)

$b = 25''$ (wing span)

Trunnion location at $0.20\mathrm{c}_{\mathrm{mean}}$

1/16 scale model

Airfoil section: NACA 23016.5 at the extended root and an NACA 23012 at the tip with a straight taper